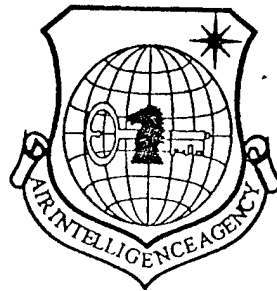


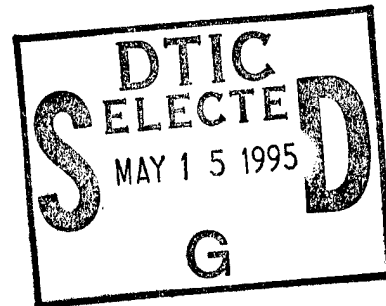
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THERMAL DESIGN OF RETURNABLE SATELLITES

by

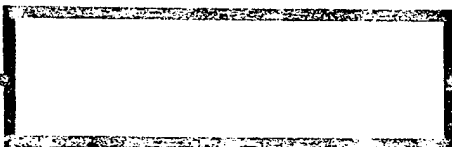
Xu Jiwan



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Xu Jiwan

ABSTRACT

This introduces China's returnable satellite thermal control design plans. On the basis of the special characteristics of China's returnable type satellites, it describes the applications of thermal inertia methods in thermal design. Comparisons between ground heat experiment data and flight telemetry temperature data clearly show that the two agree together well. Heat control system operations were normal. Satellite body and satellite interior instrument temperatures all fell relatively well into required ranges.

SUBJECT TERMS: Recoverable satellite, Temperature control, Thermodynamic test, Thermal analysis, Design

I. FORWARD

As far as the objectives and contents associated with returnable satellite thermal design are concerned, they are--through opting for the use of various types of thermal control measures--to supply for satellite instruments and equipment a beneficial thermal environment, making them capable of operating normally, and, in conjunction with this, to place them into the environmental temperature ranges required for optimum properties.

Normally, satellite thermal designs include three basic areas:

(1) Research and analyze heat sources, heat sinks, and heat exchange paths which satellites can meet in orbit. Set up thermal numerical models, and carry out thermal analyses.

(2) Opt for the use of appropriate thermal control measures, maintaining instruments and equipment on satellites at required temperature levels and within temperature ranges.

* Numbers in margins indicate foreign pagination.
Commas in numbers indicate decimals.

(3) Empirical verification of ground thermal tests.

As far as Chinese returnable satellites are concerned, exterior forms are circular cones with spherical heads. Main satellite bodies are divided into two large parts--the returnable cabin and the instrument cabin. Within the returnable cabin, instruments are mounted on cross beams. The instrument cabin is again divided into the transitional section and the sealed off cabin. Within the sealed off cabin, instruments are mounted on large beams.

The key characteristics of returnable type satellite thermal control design are as follows:

(1) Outside the satellite there is no cowling cover. In the ascent flight stage, the satellite surface will experience relatively strong aerodynamic heating and gas flow erosion environments.

(2) With regard to satellites moving in near Earth orbits, orbital periods are short. In maximum shadow orbit, satellite shadow areas occupy a large proportion of entire orbital periods, reaching 41%. Moreover, Earth light reflection and Earth infrared radiation also account for a large proportion of heat flow received by satellites other than from space.

(3) Periods of satellite motion in orbit are short.

(4) In the case of satellite three-axis stability, one surface is fixed facing the ground from beginning to end.

(5) Satellite structures are complicated. Heat capacities are relatively large.

(6) In orbit, the interiors of satellite sealed off cabins are placed in gas filled configurations. /50

(7) Entire satellites make use of chemical batteries as power sources.

(8) Within satellites, the largest part of instrumentation and equipment has relatively high requirements with regard to ambient temperatures.

Due to the characteristics discussed above, returnable type satellite heat control designs opt for the use of passive heat control as the main plan. The thermal control structure schematic is as shown in Fig.1.

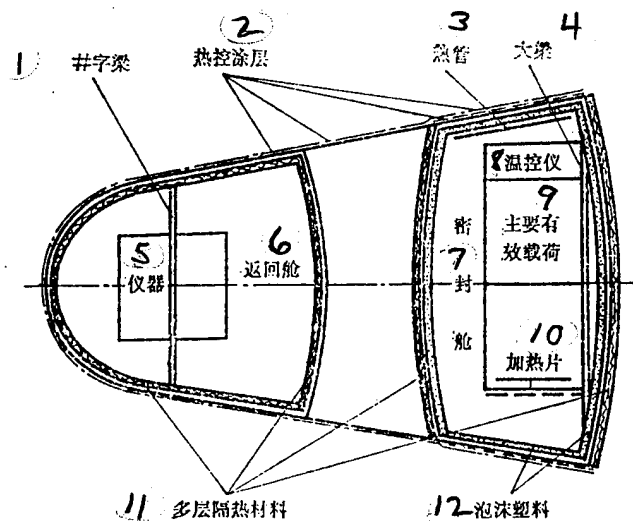


Fig.1 Returnable Type Satellite Heat Control Structure Schematic

Key: (1) Cross Beam (2) Thermal Control Coating (3) Heat Tube (4) Large Beam (5) Instruments (6) Returnable Cabin (7) Sealed Off Cabin (8) Temperature Control Instruments (9) Primary Effective Load (10) Heater Plate (11) Multilayer Heat Insulation Material (12) Foam Plastic

II. HEAT ANALYSIS AND HEAT CONTROL PLANS

As far as heat analyses associated with returnable type satellites are concerned, the first step is to set up thermal numerical models. On the basis of the overall satellite layout, structure, orbit, and attitude parameters, thermal characteristics between various parts of satellites are divided into a certain number of isothermal nodal points, acting as a simplified numerical model for thermal analysis calculations.

On the basis of flight mission requirements, returnable type satellite attitudes are triaxially stable. They are fixed facing

the ground. It is possible for satellites to meet with relatively large changes in included angles between the surface of orbital motion and sunlight. Satellites are effected by relatively large changes in sunlight exposure factors $\tau_s/\tau_0 = 0.59 - 0.75$. The result is that, in the case of different orbital motions, the differences in heat flows reaching satellite surfaces are relatively large. With regard to maximum shade orbits, sunlight exposure factors $\tau_s/\tau_0 = 0.59$. At this time, orbital surfaces and sunlight are close to parallel. Within the same cycle, space heat flows received by different locations on satellites as well as by the same locations at different times all show very large differences. Concerning orbits with sunlight exposure factors $\tau_s/\tau_0 = 0.75$, in sunlight areas, half the satellite body always receives sunlight irradiation. However, the other half sees no sunlight from beginning to end, creating huge differences associated with the heat flows reaching illuminated surfaces and surfaces away from the sun.

Under different orbital conditions, in order to solve for space heat flows absorbed by different satellite locations as well as satellite body surface temperature distributions, on satellite conical surfaces along a peripheral direction by cabin sections, respectively, there are divisions into a number of individual isothermal nodal points. With the addition of cone and lower seal tip isothermal nodal points, the whole satellite body is divided altogether into over one hundred individual isothermal nodal points. The energy equilibrium formula associated with isothermal nodal points is:

$$Q_{se} + Q_i + Q_c + GC_p \frac{dT}{d\tau} = \sigma \epsilon \bar{T}^4 \quad (1)$$

In the equation, Q_{SE} ---the sum of solar radiation, global reflection, and globally radiated heat flows associated with isothermal nodal point surface absorption;

Q_I ---generated thermal power within isothermal nodal points

Q_C ---heat exchange between isothermal nodal points

G ---isothermal nodal point mass /51

C_p ---isothermal nodal point mean specific heat

$d\bar{T}/d\tau$ ---isothermal nodal point mean temperature change speed

σ ---Stefan-Boltzmann constant

$\bar{\epsilon}$ ---isothermal nodal point surface mean infrared emission rate

\bar{T} ---isothermal nodal point surface mean temperature.

On the basis of thermal analysis calculation results, determining satellite thermal control system plans is the fact that the whole satellite takes passive thermal controls as its foundation, supplementing them with active thermal controls with regard to instruments and equipment having special requirements. We will now respectively provide explanations.

1. Returnable Cabin

Due to the fact that cabin walls will experience strong aerodynamic heating, cabin shell structure opts for the use of nonmetallic thick wall ablative materials. When satellites are in motion in orbits with maximum sunlight exposure factor $\tau_s/\tau_0 = 0.75$, as far as high temperatures on sides toward the sun and low temperatures on sides away from the sun are concerned, with regard to certain ablative material structures, there is a

possibility of this leading to severe satellite body structure deformations, even to the point of rupture. In order to lower satellite body surface peripheral direction temperature differences, it is required, on sides toward the sun, to opt for the use of coatings with low specific solar absorption rate values α_s with regard to infrared emission rates ϵ_H . However, due to the fact that, within returnable cabins, generated thermal powers are only 9W, in maximum shadow orbits, with sunlight exposure factors $\tau_s/\tau_0 = 0.59$, cabin interior temperature environments are kept above 0°C. There is also a requirement to raise cabin outer surface mean temperatures. This, then, requires opting for the use of relatively high absorption-radiation specific value (α_s/ϵ_H) coatings. At the same time, in order to reduce the amplitude of surface temperature fluctuations between sunlight areas and shadow areas in orbit, it is also necessary to opt for the use of low infrared emission rate (ϵ_H) coatings. Finally, on the basis of thermal calculation optimization compromise results, during situations of maximum sunlight exposure factor $\tau_s/\tau_0 = 0.75$, satellite surfaces facing the sun directly irradiated by sunlight account for a zone 3/20th the peripheral cabin surface area. Opting for the use of coatings associated with slightly lower absorption-radiation specific values ($\alpha_s/\epsilon_H = 1.0$), and, on remaining surfaces, opting for the use of coatings associated with slightly higher absorption-radiation specific values ($\alpha_s/\epsilon_H = 1.2$), the infrared emission rates of the two types of coatings are both $\epsilon_H = 0.5$. Thermal analysis clearly shows that this type of thermal design, in situations of high and low temperatures associated with maximum sunlight exposure factors and maximum shadow orbits, in all cases, is capable of satisfying cabin wall temperature differential and cabin interior instrument and equipment temperature environment requirements. In returnable cabin thermal designs, considering high temperature situations associated with maximum sunlight exposure factors, in cabins, temperature differences between locations close to sun irradiated

surfaces and locations on surfaces away from the sun are relatively large, creating relatively large temperature differentials between instruments. It is not easy to satisfy instrument temperature requirements. As a result, multilayer heat insulation materials are installed on returnable cabin interior walls, thus evening up cabin interior peripheral temperatures. After opting for the use of this type of measure, during satellite reentry, it also has the effect of controlling cabin interior instrument temperature rises. Again, on the basis of the characteristic of returnable cabin interior heat generation powers being small, in order to maintain interior cabin temperature levels in maximum shadow situations--between cross beams and cabin walls--option is made for connection methods associated with heat insulation blocks and heat insulation screws in order to increase thermal resistance. Besides opting for the use of the measures described above, use is also made of ground temperature adjustment systems, aiding returnable cabin thermal inertia and controlling entry track temperatures in order to guarantee returnable cabin temperature requirements during orbital motion.

2. Equipment Cabin

Due to cabin walls being thin walled metallic skin structures, cabin exteriors have no cowling covers. On the basis of thermal calculations, in order to isolate the influences of reducing and increasing sections of aerodynamic heating on sealed cabin metallic skin structure and cabin interior instrument temperatures, before painting heat control coatings on sealed cabin outer surfaces, a certain thickness of thermal insulation coating is sprayed on beforehand in order to increase the thermal resistances associated with cabin bodies and outer boundary environments. The design of thermal control coatings outside heat insulation coatings is the same as returnable cabins. On the basis of coating optimization design results, option is made

for the use of composite coatings. In situations with maximum sunlight exposure factors, direct sunlight irradiated surface locations account for 4/20 of cabin conical surfaces. Coatings associated with relatively low absorption-radiation specific values ($\alpha_s/\epsilon_H = 0.35$) are sprayed on. However, on remaining surfaces, coatings associated with relatively high absorption-radiation specific values are sprayed on in order to control overall cabin temperature levels and reduce the amplitude of cabin interior temperature fluctuations associated with high and low temperature operating conditions. In order to adjust thermal resistances at different cabin interior locations to the space environment, on surface portions of cabin interior walls, different thicknesses of foam plastic (10-30mm) are glued on. Following this, on the whole inner surface of foam plastics as well as cabin interior wall surfaces where foam plastic was not glued, in all cases, a layer of dual surface aluminum plated thin film was glued on. When determining thermal design plans, consideration was given to the fact that sealed off cabins in orbit are in a gas filled, pressurized state. Foam plastic heat insulation properties and multilayer heat insulation material are equivalent. However, its density is much smaller than multilayer material. Therefore, in sealed off cabin interiors, as far as locations where heat insulation measures must be selected are concerned, in all cases, the use of foam plastics replaced the commonly used multilayer heat insulation material. After selecting this type of measure, thermal control mass was reduced, material costs fell, and installation techniques were simplified. On the outer surfaces of sealed cabin upper and lower sealing covers, in order to isolate or reduce the influences of the outside environment, they were separately wrapped in multilayer heat insulation material. Inside cabins, instruments producing small amounts of heat or not generating heat, separately opt for the use of heat insulation cushions, glued foam plastic and aluminum plated thin film, or appropriate heat insulation wrapping layers, or placement of heat conduction plates to

conduct amounts of heat associated with instruments having large heat generation power losses to relatively low temperature instruments, and other similar measures. With regard to instruments having high heat generation power densities, black anode oxidized surfaces are required. In conjunction with this, installation surfaces are filled with heat conducting silicon grease. Going through these treatments, the temperatures of instruments and equipment at different locations inside cabins are controlled.

3. Instruments and Equipment

With regard to instruments and equipment inside cabins with special heat control requirements, it is necessary to select appropriate measures in order to satisfy the temperature requirements.

(1) Principal Effective Load Thermal Design. Key components associated with principal effective load thermal design require controlling constant temperatures. As a result, besides establishing methods of guaranteeing constant temperature control precision, there is also a requirement to lower as much as possible heat leakage in order to lower thermal control power consumption.

In order to reduce heat leakage, it is necessary to reduce the thermal conduction area between the components in question and the craft frame. Between various metallic contact surfaces, there were respectively added 1-5mm glass fiber reinforced plastic thermal insulation spacers. As far as thermal protective covers are concerned, outer surfaces were wrapped in multilayer thermal insulation material, and so on. In order to even up the temperatures of the components in question, it is necessary to strengthen radiated heat exchange, requiring black anode oxidation treatment of various metallic surfaces within

components. Effective load active thermal control measures include temperature control heating plates, heat sensitive electrical resistances, and control circuits.

(2) Battery Thermal Design. Key power sources on satellites are chemical batteries. Their characteristics are: relatively great differences between the dimensions and the masses of the various batteries; various different thermal capacities; and various different heat generation powers, installation positions, and ambient temperature conditions for the various batteries. As a result, it is necessary to carry out thermal designs for the various batteries separately.

In order to obtain heating power data for batteries in various phases, specialized measurements were carried out on heat generation amounts for batteries under different operating conditions. In conjunction with this, it was used as a basis for thermal designs.

On the foundation of battery heat generation powers already accurately known, use was made of thermal equilibrium formulae to solve for different passive thermal control measures capable of guaranteeing appropriate operating temperatures for various batteries under high and low temperature satellite operating conditions. These passive thermal control measures include: gluing foam plastic material and dual surface aluminum plated thin films on battery surfaces, the addition of glass fiber reinforced plastic spacers on installation surfaces, and so on. In the case of batteries with relatively small average heat generation powers as well as being placed in relatively low ambient temperatures, on the foundation of passive thermal control, in accordance with thermal calculation results, appropriate heating plates are set in battery interiors--being electrified and heating after entering orbit--in order to satisfy the temperature requirements.

(3) Infrared Horizon Instrument Thermal Design. Infrared horizon instruments are installed on tail sections outside satellites. Directly exposed to the space environment, they receive solar radiation, global reflection, and global radiation heat flows, giving rise to radiation and thermal conductance exchange with satellite bodies. Interiors also have small amounts of generated heat power.

On the basis of thermal analysis results, infrared horizon instrument thermal designs are primarily heat insulation designs to guarantee temperatures. This being the case, the outer surfaces are wrapped in multilayer heat insulation material, and, in addition, appropriate heat control coatings are painted on the outer surfaces of multilayer heat insulation materials. Besides this, heating plates are also installed inside infrared horizon instruments--powered up and heating in orbit--in order to satisfy the temperature requirements.

(4) Sealed Cabin Large Cabin Door Thermal Design. Sealed cabin large cabin doors run along the circular cone periphery for 92° (see Fig.2), stretching across two quadrants. During operating conditions with sunlight exposure factor $\tau_s/\tau_c = 0.75.$, one side of cabin doors is always receiving solar irradiation. Temperatures are relatively high. The other side of cabin doors, from beginning to end, does not see sunlight. Moreover, Earth reflection and infrared radiations which are received are also very weak. Temperatures are relatively low. In this way, it is possible, on large cabin doors, along circular cone peripheral directions, to create relatively large temperature differential deformations, causing large compartment door seals to lose their effectiveness. On the basis of thermal analysis results, on upper and lower large cabin door frames along circular cone peripheral directions, two aluminum heat tube channels are respectively installed in order to even up large cabin door frame temperatures so as to satisfy large cabin

door periphery temperature differential requirements.

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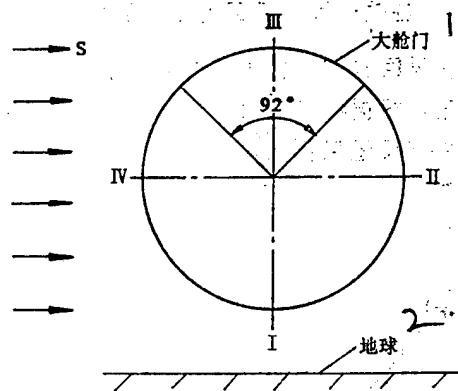
(5) Ground Temperature Adjustment Systems. In order to guarantee satellite temperature control in an erected awaiting launch configuration as well as controlling initial satellite orbit entry temperatures, ground temperature adjustment systems were designed. Ground temperature adjustment system cooling gas tubing, heating plates, and temperature sensors are respectively placed in returnable cabins and sealed off cabins.

III. APPLICATIONS OF THERMAL INERTIA METHODS IN THERMAL DESIGN

Returnable type satellite masses are large. Orbital motion times are short. There are no cowlings, and powers inside returnable cabins are small. As far as such characteristics as these are concerned, they are most appropriate for applications of thermal inertia methods.

Fig.2 Locations of the Sun and the Earth Relative to Sealed Cabin Large Cabin Doors

Key: (1) Large Cabin Door
(2) Earth



Thermal inertia methods in thermal designs are nothing else than methods applying heat capacities possessed by satellites -- during changes in ambient temperature conditions -- and satellite temperature change lag phenomena, to make instruments and equipment on satellites, within a fixed time interval, still hold within required temperature ranges.

In order to calculate -- under ambient heat flow conditions or when internal heat sources give rise to changes -- satellite surface temperature variations, take equation (1) and write it to be:

$$\frac{dT}{dt} = (\sigma \epsilon T^4 - Q_{sx} - Q_i - Q_c) \frac{1}{GC_p} \quad (2)$$

From formula (2), it is possible to see that, the larger satellite heat capacity (GC_p) is--when satellite thermal environment conditions or internal heat sources change--the smaller the speed of satellite temperature changes $\left(\frac{dT}{dt}\right)$ will then be, and the slower the temperature changes. Conversely, within the same time interval, the more rapid temperature changes will be. When considering satellite skin surface isothermal nodal point thermal equilibrium, in formula (2), the power quantity Q_i accounts for a very small proportion of total heat flow. It is often possible to ignore it in calculations.

Speaking in terms of cabin interior instruments, when instrument internal powers or ambient temperatures suddenly change, instrument temperature changes can be calculated using the formula below:

$$Q_p + GC_p \frac{dT}{dt} = \frac{\Delta T}{R_i} + \frac{\Delta T}{R_r} \quad (3)$$

In the equation, Q_p ---instrument heat generating power

G ---instrument mass

C_p ---instrument specific heat

dT/dt ---instrument temperature change speed

ΔT ---temperature differential between instrument

and environment

R_λ ---thermal conduction resistance between
instrument and environment

R_1 ---thermal radiation resistance between
instrument and environment

After going through arrangement, formula (3) is changed to be

$$\frac{dT}{d\tau} = \left(\frac{\Delta T}{R_\lambda} + \frac{\Delta T}{R_r} - Q_r \right) \frac{1}{GC_p} \quad (4)$$

From formula (4), it can be seen that the larger (GC_p) is, the larger are instrument and environment conduction and radiation thermal resistances. When instrument internal powers or ambient temperatures suddenly change, the smaller instrument temperature change speeds are, the slower instrument temperature changes then are. As a result, the longer the time periods then are, maintained within a fixed temperature range. Conversely, the shorter time periods then are, maintained within a fixed temperature range.

/54

Based on thermal inertia method principles, combining returnable type satellite characteristics, thermal designs were carried out for different stages of satellite operations.

1. In Orbital Phase, Applications to Returnable Cabins

Within returnable cabins, instrument heat generation power Q_I is barely 9W. In formula (1), heat exchange Q_c between returnable cabins and equipment cabins is a negative value. On the basis of antiaerodynamic heating and erosion satellite body surface coatings as well as heat insulation measures which can be selected for use in cabins, calculations clearly show that cabin interior cross beam temperatures are still below 0°C . They are not able to satisfy the largest part of instrument requirements for temperature lower limits. In order to resolve this problem, going through thermal analysis and calculations, option is made for the use of thermal inertia methods.

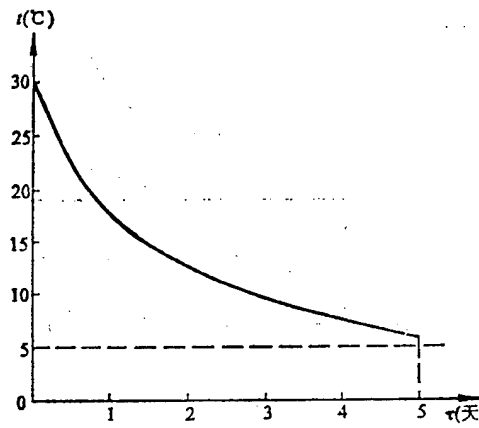


Fig.3 Relationship Curve Between Temperature and Time Within Returnable Cabins

Key: (1) Days

The concrete method of operation is this. Right before launch, through ground temperature adjustment systems, control returnable cabin cross beam temperatures, making returnable cabins, at the time of orbital insertion, store relatively large amounts of heat. In addition, within cabins, increase such various measures as thermal resistance. Despite the fact that, in maximum shadow orbits, following along with returnable cabin surface heat radiation into space, cabin interior temperatures drop over time. However, up until the final phases of satellite flight, due to thermal inertia associated with the cabins as a whole, cabin interior temperatures--from beginning to end--hold relatively well within required ranges, as is shown in Fig.3.

Operating with formula (2), it is possible to calculate crossbeam temperatures inside returnable cabins for different instants during periods of satellite flight or the periods needed for crossbeam temperatures to drop to different values.

2. In Ascent Phase, Applications to Sealed Cabins

Sealed cabin walls are thin metallic structures. In order to reduce the influences of aerodynamic heating on skin temperature increases, on sealed cabin aluminum wall outer surfaces heat insulation coatings of a certain thickness were designed, making thermal conduction resistance perpendicular to cabin surfaces increase approximately 130 fold. As a result, there were very large drops in metallic wall temperature increases, making sealed cabin metallic skin temperatures, in the ascent phase, capable of satisfying structural requirements. However, during the process of ascent, although time periods are short, on cabin walls, such heat insulation methods as gluing on foam plastics are also opted for. However, sealed cabin wall temperatures are very high, making cabin interior instrument temperatures rapidly go up. Moreover, after orbit insertion, temperature increase lags are produced, making instrument temperatures reach or go above temperature upper limit requirements. Operating with formulae (2) - (4), analytical calculations clearly show that it is possible to utilize thermal inertia methods in order to solve this problem.

Immediately before launch, option is made for the use of ground temperature adjustment systems, taking sealed cabin interior representative point temperatures and dropping them approximately 5°C. In conjunction with this, within a period of time, there is stabilization at this temperature level, making sealed cabins possess cold inertia. During the ascent phase, after aerodynamic heating and orbital insertion lag temperature increases, cabin interior instrument temperatures are still

within required ranges.

During summer satellite launches, ground ambient air temperatures are relatively high. Immediately before launch, sunlight directly irradiates satellite surfaces. Sealed cabin interior temperatures will reach 40°C. The application of thermal inertia methods in sealed cabins seems to be especially important.

If satellites are launched in fall or winter seasons, ground ambient air temperatures are at approximately 5°C or below. Immediately before launch, there is then no need in sealed cabins to apply thermal inertia methods.

3. During Ground Transport Phase, Applications to Returnable Cabins

During satellite summer launches, in ground site to site transport, although periods of time are short, vehicle casings for satellite transport, however, will receive strong solar irradiation. Vehicle casing wall temperatures sometimes rise as high as 70°C. Some instruments, after satellite loading, require storage or operation in room temperature environments. If temperature control methods are not opted for in surface transport phases, these instruments and equipment may possibly exceed required temperature ranges and lose their effectiveness. In order to resolve ground transport phase satellite interior temperature control--going through comprehensive analysis--the simplest, most economical, most reliable, and effective methods were none other than utilizations of thermal inertia methods. /55

IV. GROUND THERMAL EXPERIMENTS AND FLIGHT EXPERIMENTS

In satellite development stages--going through adequate thermal analysis and verification--after executing thermal designs, it is necessary to carry out a series of ground tests including active phase aerodynamic heating tests, aerodynamic erosion tests, vacuum heat equilibrium tests, as well as multiple iterations of flight tests, and so on.

1. Active Phase Aerodynamic Heating Tests

As far as aerodynamic heating tests are concerned, they are experiments to simulate satellite aerodynamic heating states in active phases, measuring satellite surface skin temperature increases. The tests use iodine tungsten lamps to simulate aerodynamic heating thermal flows. Cabin body structures are 1:1 structural models. Cabin surfaces, after being sprayed with the required coatings, are sprayed again with a layer of black coating. Foam plastics are glued to cabin interior walls. During tests, on the basis of given heat flow curves, iodine tungsten lamps are used to heat sealed cabin surfaces, measuring surface temperature changes.

2. Aerodynamic Erosion Tests

With regard to aerodynamic erosion tests, they use gases which possess the flight speeds and high temperatures of satellites in the ascent phase, erosion of satellite models, and tests checking changes in satellite surface thermal control coating properties.

The tests respectively carry out two types of operating states--high pressure heads blowing cold wind and low pressure heads blowing hot wind. Test results clearly show that satellite

model surface thermal control coatings are intact. Thermal properties do not change.

3. Vacuum Thermal Equilibrium Tests

In the case of vacuum thermal equilibrium tests, they are experiments which take satellites and place them in equipment simulating orbital space environments (vacuum, space thermal flows, heat sinks, and so on), using them in order to empirically verify whether or not satellite thermal designs are correct.

When determining returnable type satellite vacuum thermal equilibrium test plans, besides carrying out tests according to conventional methods, the experimental methods below are also brought up or selected for use.

(1) In orbit, sealed cabin interiors are filled with gas. On the basis of thermal analysis results, if one uses cabin interior gas filled states to carry out sealed cabin vacuum thermal equilibrium tests, natural convection heat exchange associated with the ground will make experimental results severely distorted. In orbit, gases inside cabins have no natural convection heat exchange. Cabin interior gas molecule heat conduction also accounts for only 4% of solid heat conduction and radiation heat exchange. It is possible to take this and include it in overall considerations of thermal conduction quantities. As a result, it brings up experimental methods associated with using cabin interior vacuum states in order to simulate cabin interior gas filled states in orbit. With regard to requirements put forward for degrees of cabin interior vacuum during tests, monitoring is carried out during thermal experiments, guaranteeing the authenticity of thermal tests.

(2) During prototype satellite thermal tests, it is necessary to adjust sealed cabin surface coatings. Only then is it possible to satisfy instrument temperature requirements. Option is made for the use of simulation test methods associated with unopened vacuum chamber altering of satellite surface coatings. Expected objectives were attained.

(3) During thermal experiments, in accordance with the principle that differences between satellite surface design temperatures and actually measured temperatures are proportional to changes in infrared heating cage powers, experimental methods are put forward to use computers to automatically adjust infrared heating cage powers in order to simulate space exterior heat flows. Option is made for the use of this type of method. In high and low temperature operating states, satellite thermal equilibrium errors are less than 1.25%, guaranteeing thermal test precision and data reliability.

(4) During verification satellite thermal tests, on the basis of prototype satellite thermal test results, option is made for acceleration heat experimental methods.

Prototype satellite vacuum thermal equilibrium tests were carried out on returnable type satellites. Satellites are 1:1 models. Simulation instruments are installed inside satellites. Inside them, electrical resistances are installed in order to simulate instrument heat generation powers. Infrared heating cages are used to simulate experimental space heat flows. Test results clearly show that thermal designs are basically capable of satisfying temperature ranges required by instruments on satellites. On the foundation of measures for adjusting prototype satellite partial heat controls, verification satellite vacuum thermal equilibrium tests were carried out. Test models verify all instruments actually installed within satellites and satellite bodies, simulating orbital flight status operations. Through experiments, it was discovered that chemical batteries

used on satellites had large loads as well as large amounts of generated heat. Specialized measurements were done of amounts of heat generated by batteries, adjusting partial battery heat control measures, thus, finally settling on a control technology configuration.

V. CONCLUSIONS

(1) As far as the results from numerous flights of returnable type satellites are concerned, all instrument temperatures on satellites are relatively well within design ranges. Moreover, ground thermal tests and flight test data agree very well. Flight data replicability is also very good. The explanation for this is that heat control system designs and thermal analyses are rational. Thermal test methods and thermal experimentation results are effective and correct.

(2) With regard to returnable type satellites which possess such characteristics as large heat capacities, short flight times, small returnable cabin interior heat generation powers, and so on, in different stages of satellite operation, thermal inertia methods are effective methods of thermal design. In many applications, good results have been uniformly obtained.

(3) Satellite body surface coating designs effectively control various cabin temperature levels. In the case of sealed cabin surface heat insulation coating design, it controls cabin wall temperature rises caused by aerodynamic heating, reaching design indices.

(4) In satellite main effective loads, option was made for the use of passive and active heat control technology combinations. As far as constant temperature controlled key component temperatures are concerned, during entire periods of orbital movement, operating reliability, temperature control

precision, and power consumption were all better than design indices.

(5) Realization clearly verifies that carrying out specialized measurements on amounts of heat generated by batteries, and, in conjunction with that, using them to act as the basis for battery thermal design is extraordinarily necessary.

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